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GEORGE C. MARSHALL SPACE FLIGHT CENTER
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TEMPERATURE CONTROL OF THE S-30 PAYLOAD (EXPLORER VIII)

By

William C. Snoddy

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ABSTRACT

This report concerns the thermal design of the S-30 payload (Explorer VIII, Ionosphere Probe Satellite) placed into orbit by the 19-D Juno II vehicle on November 3, 1961. The discussions include:

(1) need for thermal design

(2) approach used

(3) theoretical model

(4) design tests, and

(5) comparison of actual in-orbit data with theoretical data.

It is concluded that the design was successful, due to the fact that with a desirable temperature range of 0° to 50°C, the instrument column remained between 22°C and 33°C, and the batteries between 17°C and 27°C.

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SPACE THERMODYNAMICS BRANCH
RESEARCH PROJECTS DIVISION

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SUMMARY

This report concerns the thermal design of the S-30 payload (Explorer VIII, Ionosphere Probe Satellite) placed into orbit by the 19-D Juno II vehicle on November 3, 1961. The discussions include:

(1) need for thermal design

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INTRODUCTION

The NASA S-30 payload (Explorer VIII) was known as the Ionosphere Direct Measurements Satellite and had as its major objective the study of temporal and spatial distribution of ionospheric parameters at altitudes between 320 and 1280 kilometers (200 and 800 nautical miles) [1] It was placed in orbit November 3, 1960 using the Juno II-19D vehicle. The payload was powered by mercury batteries and had an operational lifetime of about two months.

The need for controlling the temperature of this payload arose from the thermal sensitivity of the instrumentation and batteries. It was required that the payload be designed such that the temperature of these components would remain within the range of 0° to 50° C.

The temperature of an object in a vacuum such as space is determined by the radiation exchange with its environment and any internal energy conversion to or from heat. In the case of a near-earth satellite, the major radiation environment is made up of direct solar radiation, reflected solar radiation from the earth, earth infrared radiation, and reflected and infrared radiation from the satellite. Internal energy conversion to or from heat in most satellites thus far has been limited to ohmic heat generation.

The amount of heat entering a satellite varies throughout a revolution as the payload passes through the earth's shadow, changes altitude and earth-fixed attitude, and varies in angular distance from the subsolar point on the earth. A much slower effect on heat input occurs with changes in the plane of the orbit with respect to the sun and the earth, the argument of perigee, the percent of the orbit in sunlight, and the solar attitude of the payload. The amount of heat leaving the payload varies solely as the fourth power of the temperature of the radiating surface. This is under the assumption that there are no changes in the amount of external area or the surface characteristics of this area.

The internal heat generation for the S-30 payload was small. Almost all of this internal heat was generated in the area of the transmitter and caused only a few degrees rise in the temperature of the nearby instrumentation.

THERMAL DESIGN PHILOSOPHY

From preliminary studies, it was determined that the passive methods of temperature control used on earlier Explorer satellites would be sufficient for this payload [2, 3, 4]. This meant that while the temperature of the skin of the payload fluctuated widely during a revolution as the amount of heat entering the payload changed, the time-average temperature was controlled by proper selection of the radiative characteristics of the surfaces. The temperature fluctuations were damped out from the thermally sensitive components by the use of insulation. The eight battery packs were covered with highly reflective aluminum foil to prevent radiation and mounted on Kel-F spacers to prevent conduction. The instrument column was supported by fiberglass spacers inside an aluminum cylinder, whose inner surface was highly polished (Figs. 1 & 2).

The insulation damped out most of the fluctuations during one revolution, but was not sufficient to damp out the variations in the average value of the skin temperature over a period of several days. Consequently, there was a slow day-to-day variation in the temperature of the sensitive components.

The proper radiation characteristics of the external surfaces, which were all aluminum, were obtained by use of a "sandblasting" process and two types of paint. The "sandblasting" process was originally intended to be the only surface treatment, however, emissivity measurements made on samples attached to the surfaces of the flight payloads during this process indicated that additional corrections were necessary. This was accomplished using a red, iron oxide paint and an electrically conductive silver paint. The red paint had a solar absorptivity (α) of 0.73 and an infrared emissivity (ε) of 0.80, while the silver paint had an α of 0.33 and an ε of 0.47. Since the red paint was not a conductor of electricity, it had to be applied in a pattern conducive to the maintenance of an equipotential surface (Fig. 3). This was essential since the

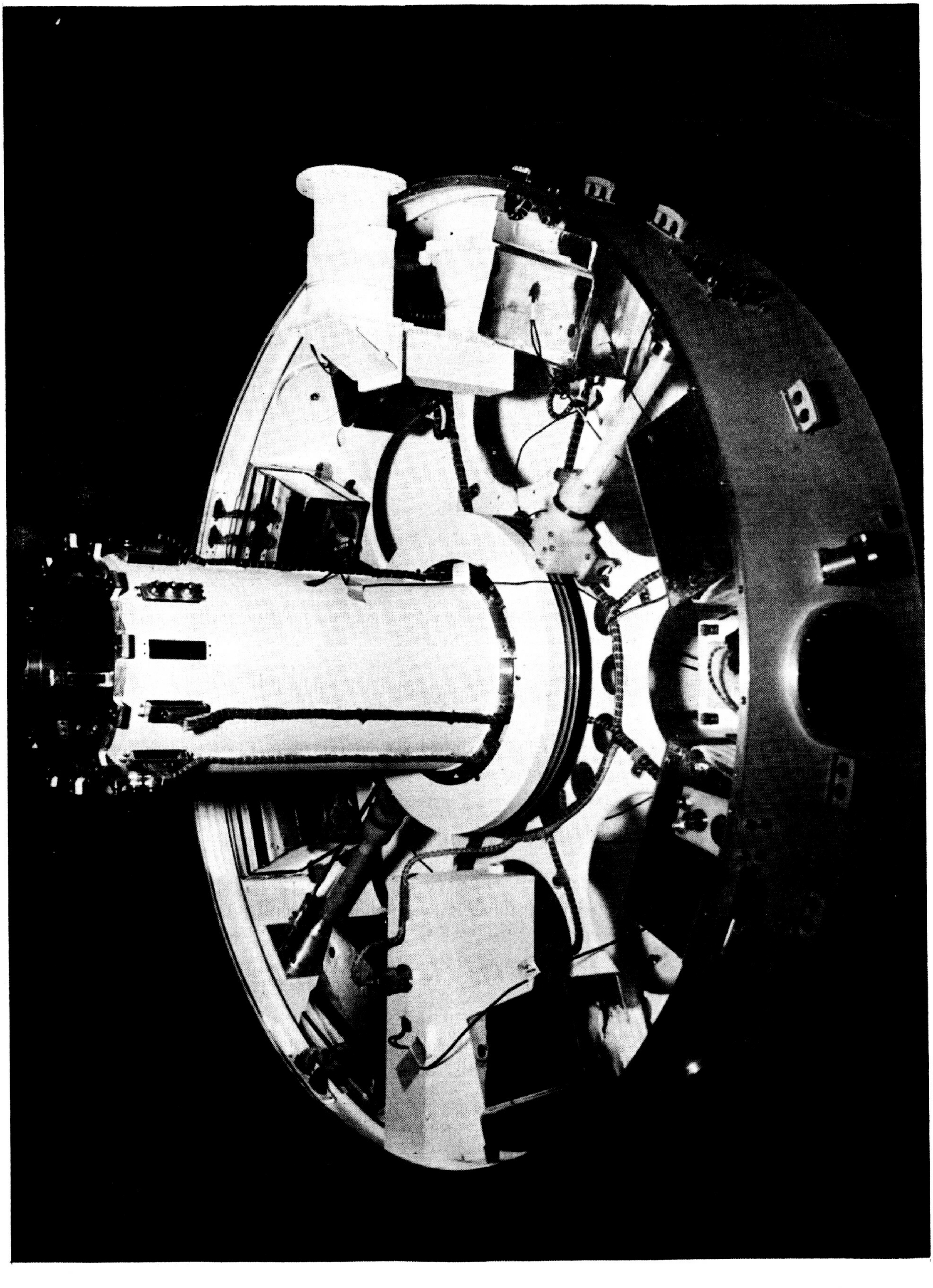


FIGURE 1. INTERIOR VIEW OF THE PAYLOAD

FIGURE 2. INSERTING THE INSTRUMENT COLUMN

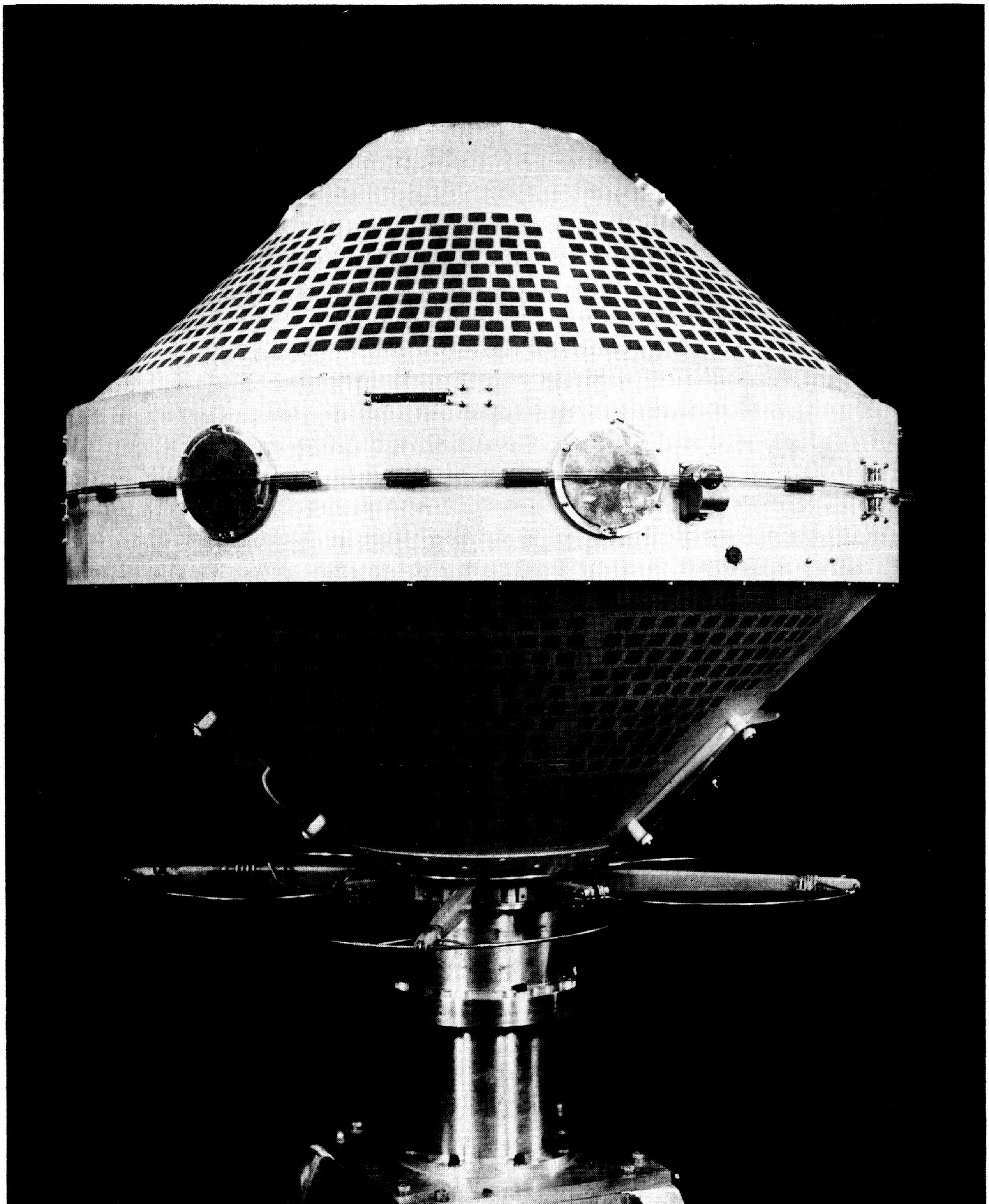


FIGURE 3. EXTERNAL VIEW SHOWING THE THERMAL COATINGS

purpose of the satellite was to measure ionospheric parameters directly, and thus, any effect the presence of the satellite had on the ionosphere had to be minimized. Table I provides a breakdown of surface coatings used on the various areas of the satellite.

Besides insulating the batteries and instrument column from the skin, additional internal aspects of the thermal design included painting the inside of the skin with a white TiO paint. This paint has a high infrared emissivity and, thus, to a great extent, allows the surfaces to equalize in temperature. This was done in order that no "hot" or "cold" spots would develop and to prevent any extremely high thermal gradients across the instrument column.

THEORETICAL CALCULATIONS

A theoretical temperature program was set up on the IBM 7090 computer as an analytical aid in completing the thermal design of the payload. The satellite was broken up into 11 areas, each assumed to be isothermal. A heat balance equation of the following form was prepared for each of the areas:

$$T_i H_i = A_{1i} \alpha_i S D_1 + A_{2i} \alpha_i B S g D_2 + A_{3i} \varepsilon_i E S g - A_{4i} \varepsilon_i \sigma T_i^4$$

$$+ \sum_j \left[R_{ij} (T_j^4 - T_i^4) + C_{ij} (T_j - T_i) \right]$$

where

T_i = temperature of area "i"

H_i = heat capacity

A_{1i} = effective area to solar radiation

α_i = solar absorptivity

S = solar radiation flux

D_1 = 1 when satellite is in sunlight
0 when satellite is in shadow

A_{2i} = effective area to earth albedo

B = ratio of earth albedo flux to solar radiation flux

g = altitude factor

D_2 = factor based on angular distance of satellite from sub-solar point on earth.

A_{3i} = effective area to earth infrared radiation

E = ratio of earth infrared flux to solar radiation flux

A_{4i} = radiating area

σ = Boltzmann constant

R_{ij} = radiation constant between areas "i" and "j"

C_{ij} = conduction constant between areas "i" and "j"

T_j = temperature of area "j"

Q = internal heat generation

ε = infrared emissivity

This group of nonlinear differential equations was then numerically solved throughout the complete orbit. By using the computer program, factors such as the amount of paint and insulation needed were determined. The program is also used in evaluating the temperature data recorded from four thermisters on the satellite and the data obtained during thermal testing.

TABLE I

Breakdown of Surface Coatings used on Various Areas of Satellite

AREA	α	ε
Top Plate	0.44	0.26
81% "sandblasted" aluminum	0.50	0.32
19% gold (field sensor)	0.20	0.05
Top Cone	0.52	0.42
58% "sandblasted" aluminum	0.50	0.27
22% red paint	0.73	0.80
20% silver paint	0.33	0.47
Equator	0.51	0.32
81% "sandblasted" aluminum	0.55	0.28
19% silver paint	0.33	0.47
Bottom Cone	0.48	0.32
42% "sandblasted" aluminum	0.50	0.27
40% silver paint	0.33	0.47
18% red paint	0.73	0.80
Bottom Plate	0.50	0.32
100% "sandblasted" aluminum	0.50	0.32
Micrometeorite Sounding Boards	0.20	0.05
100% polished aluminum	0.20	0.05

THERMAL TESTING

A prototype of the payload was subjected to two types of thermal tests in order to check the various thermal transfer coefficients and thermal time constants. One test consisted of cooling the prototype to a uniform temperature of about -20° C. Then, while in a vacuum of 5×10^{-4} mm Hg, or better, the outside surfaces were raised to a temperature of 70° C and the temperatures of various parts of the instrument package were recorded. From these tests, it was possible to get the over-all thermal transfer coefficients between various parts of the instrument package and the skin.

In the second set of tests, an actual orbital situation was simulated. In orbit, the payload spins about a longitudinal axis passing through the length of the instrument column. The spinning motion would equalize the temperature around the payload, but would not reduce the gradient from top to bottom. Therefore, a "worst situation" condition was simulated where the sun would be "seeing" the payload at an angle parallel to the spin axis. The payload was again cooled to a uniform temperature of about -20° C. Then, while under vacuum, heat was applied to those areas that the sun would "see" at this solar attitude angle. By use of the theoretical temperature computer program, calculations were made to determine the net amount of heat to apply to these surfaces as a function of their temperatures. As the surface temperatures increased, the heat inputs were decreased to the point where the heat entering the surfaces no longer increased the temperatures of the surfaces, but was conducted and radiated to the unheated parts of the prototype. These unheated surfaces were allowed to radiate as freely as possible. Thus, a steady state condition, very close to the condition in space previously described, was established. Gradients were established between various parts of the satellite, and with a knowledge of the heat input, the thermal transfer coefficients were determined.

The tests were conducted in the same manner as were the thermal tests on the S-1 and S-46 payloads [5, 6]. The test requirements were specified to the Electro-Mechanical Branch of Guidance and Control Division and made a part of the test specifications. A detailed procedure was set up by the E-M Branch to use the same method of blanket heating and liquid nitrogen cooling as used in the earlier thermal tests.

The vacuum requirement was met by running the tests in a large chamber capable of attaining a pressure of about 3×10^{-7} mm Hg. The chamber and associated equipment are shown in Fig. 4.

The payload was initially cooled by a special cone made of stainless steel tubing completely filled with liquid nitrogen (Figs. 5 and 6). The same cone was used during the test as a radiation heat sink. Since the inside surface temperature of the cone was that of liquid nitrogen, it radiated very little heat to the payload; by coating this inside surface with a special black carbon paint, the heat radiated by the payload was almost entirely absorbed. This combination of cold temperature and high absorptivity gave a good simulation of the perfect radiation heat sink of space.

During the cooling-down process, the vacuum chamber was filled with dry nitrogen so that cooling could be accomplished with the aid of convection. When the temperatures equalized at about -20° C, the dry nitrogen was pumped out and heat applied using specially designed fitted blankets (Figs. 7 and 8). During the orbital simulation tests, the actual heat input was controlled by measuring the blanket current and voltage, assuming that all of this ohmic heat went into the payload by use of foil reflectors (Figs. 9 and 10). During these tests, up to 27 thermocouples were attached to the payload, heater blankets and heat sink to record the temperatures. Using a stepping switch, the thermocouple readings were recorded on strip charts in the control room where the controls for the blanket circuits were located.

After the blankets were turned on, the temperature measurements were taken at the rate of one reading every half minute for the first hour, every 5 minutes for the next 3 hours, and every 10 minutes for the remainder of the test. On the orbital simulation tests, the blanket energy input data was taken at the same rate. Pressure readings were made commensurate with noted fluctuations. The location of the thermocouples are given in Table II.

Some of the results from one of the "full blanket" tests where heat was applied to all external surfaces are given in Fig. 11. The sharp rise in temperature imposed on the shell of the prototype is shown by the curve labeled TC-2. The insulation of the batteries (TC-7) and instrument column (TC-19) from the shell is evident by their slow response

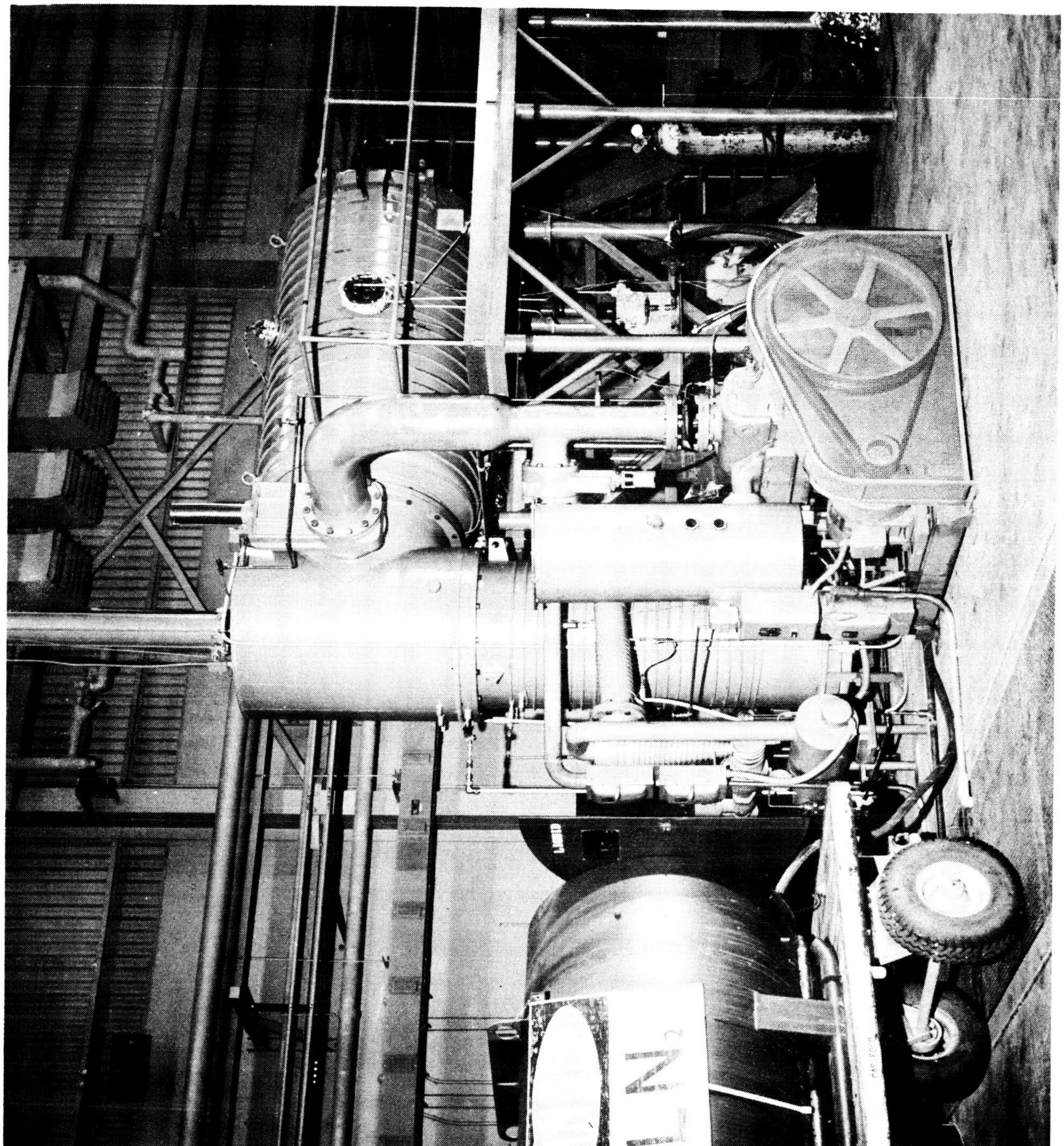


FIGURE 4. VACUUM CHAMBER

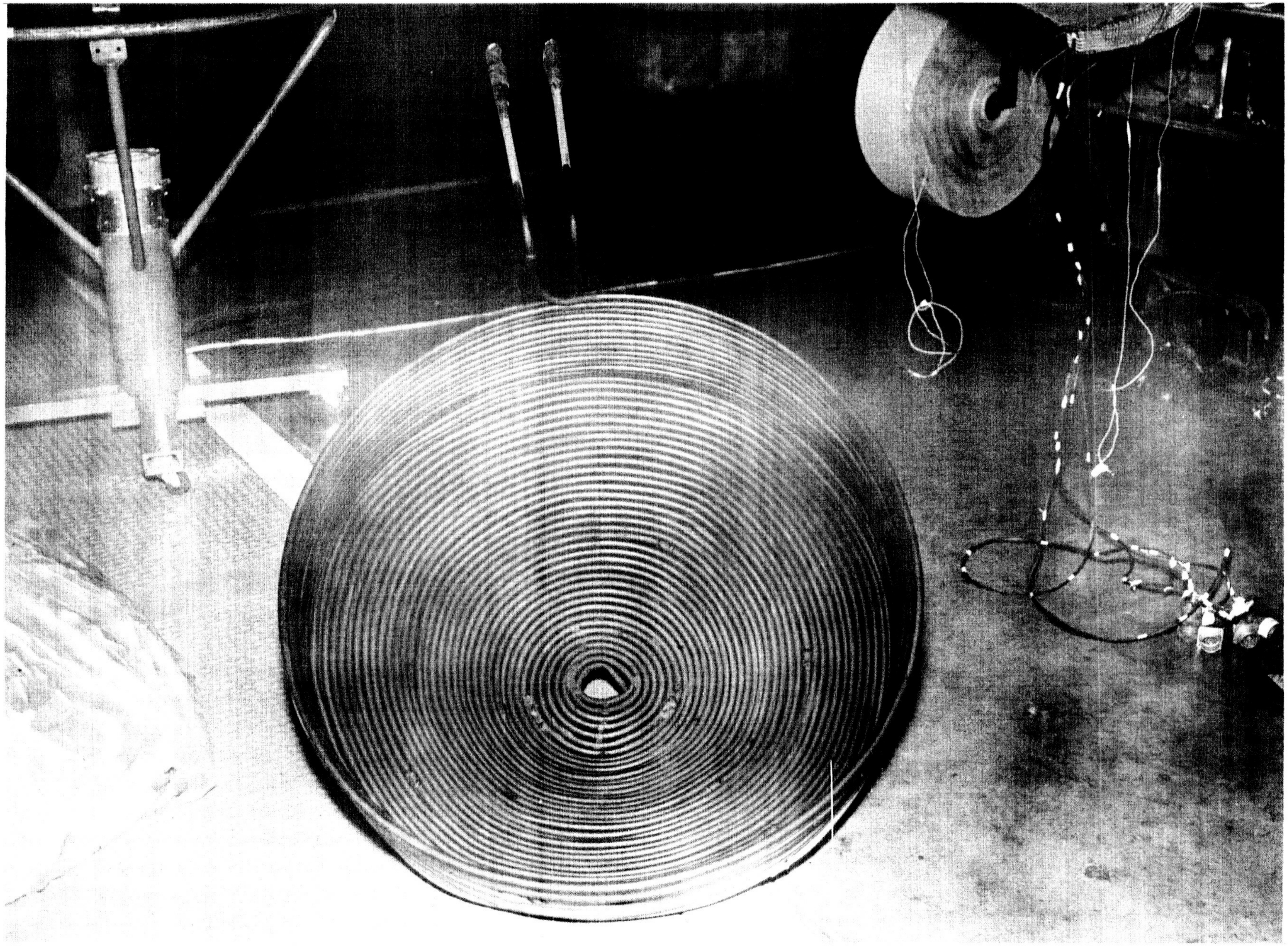


FIGURE 5. LIQUID NITROGEN CONE

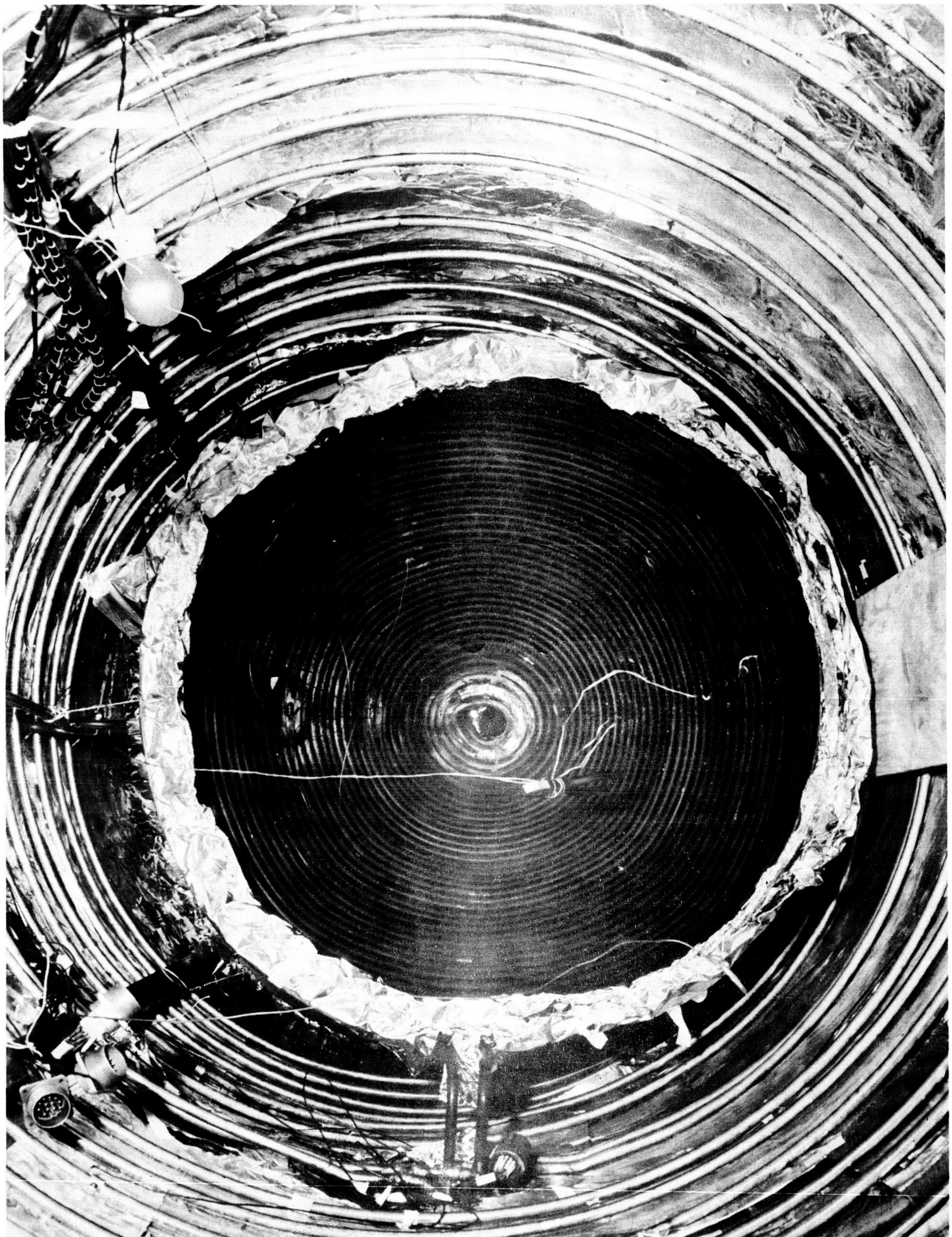


FIGURE 6. LIQUID NITROGEN CONE IN THE VACUUM CHAMBER

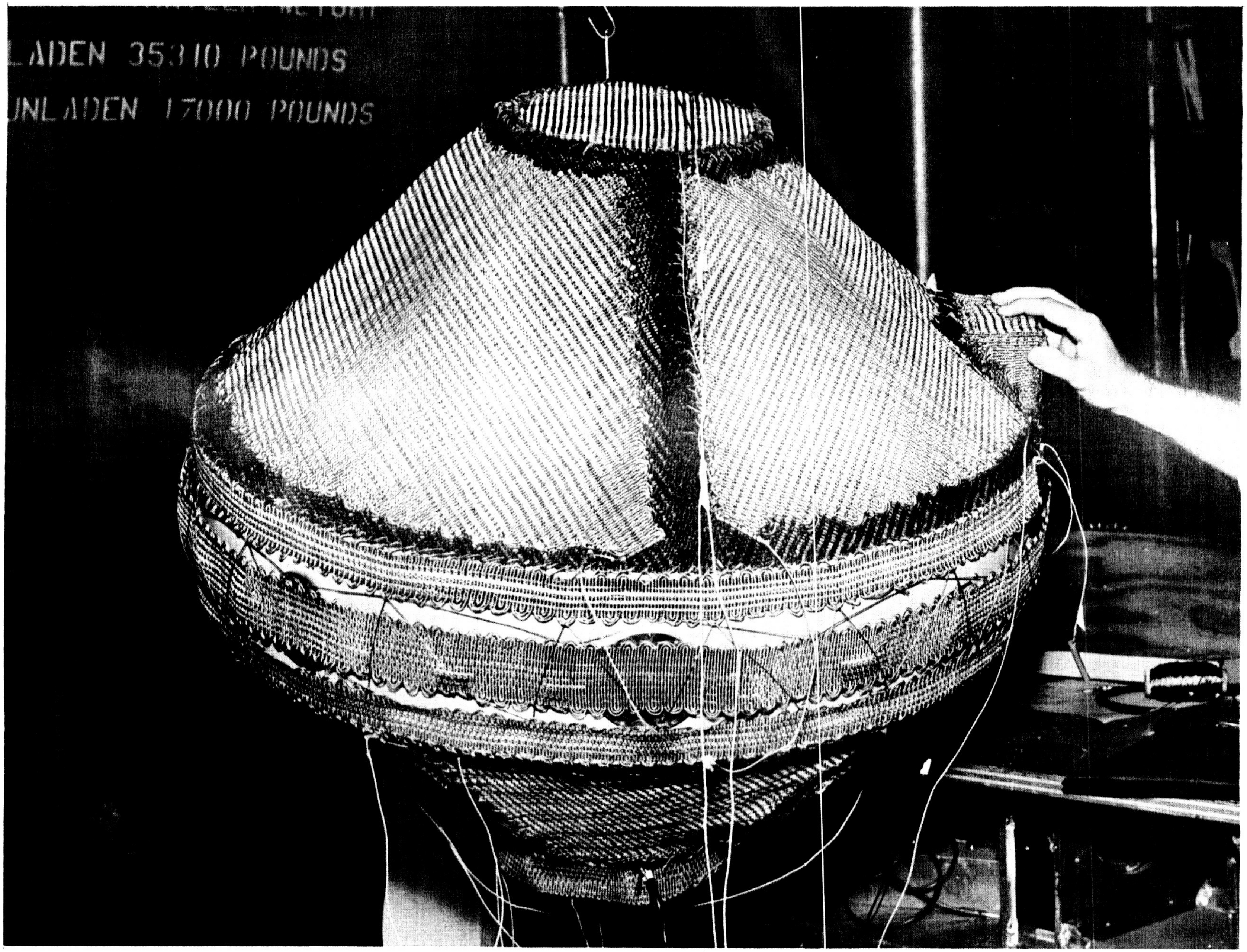


FIGURE 7. FULL BLANKET ON THE PROTOTYPE

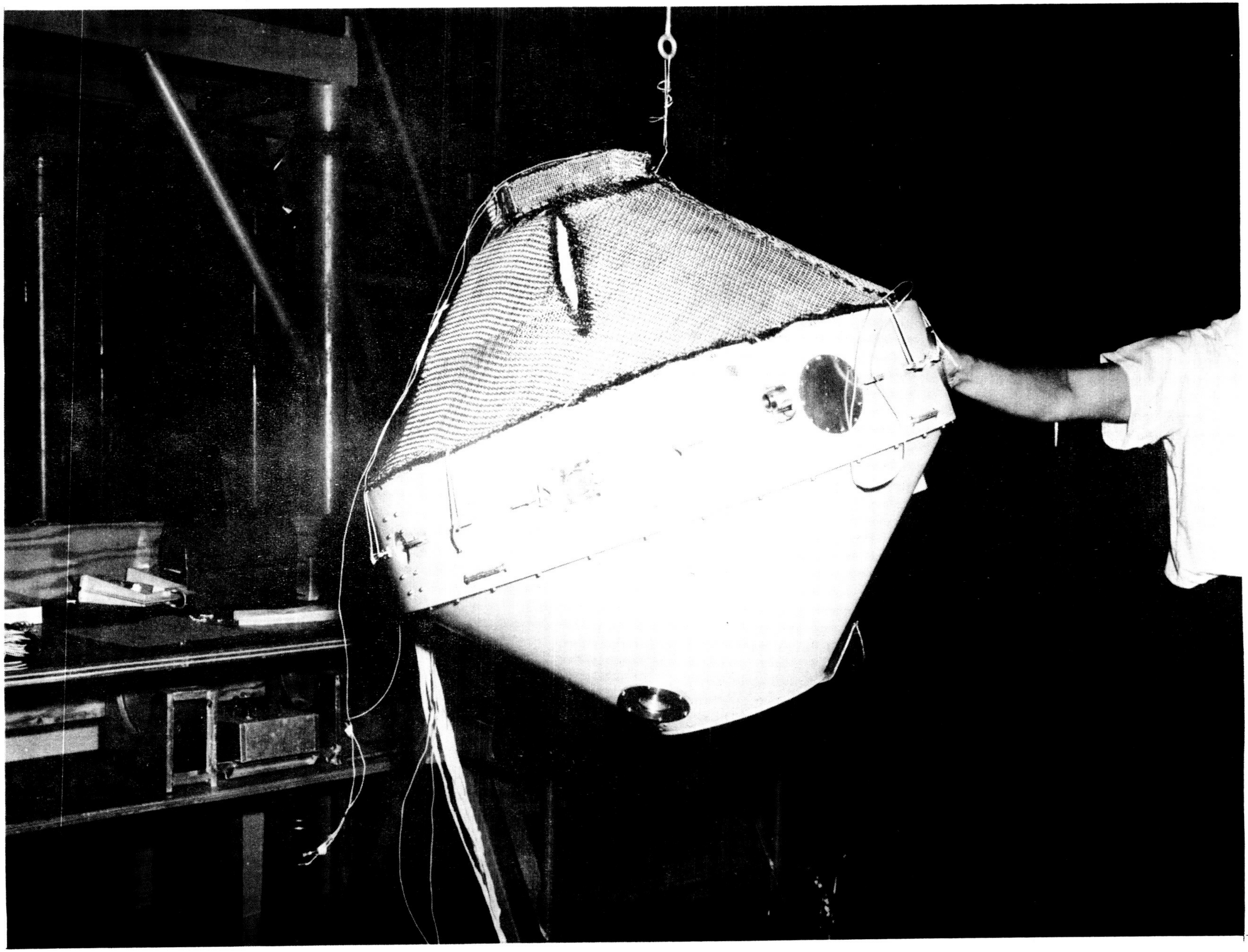


FIGURE 8. HALF-BLANKET ON THE PROTOTYPE

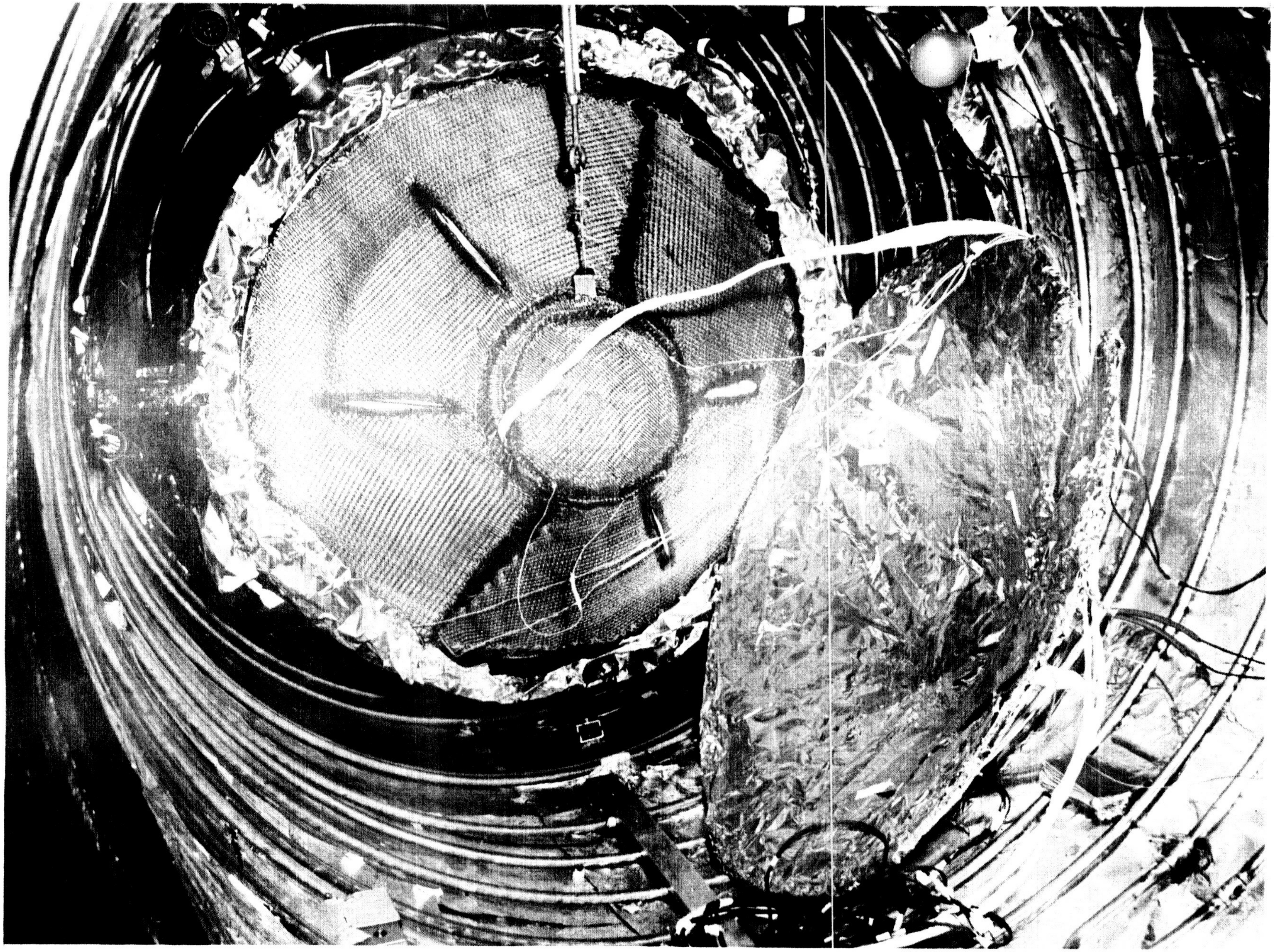
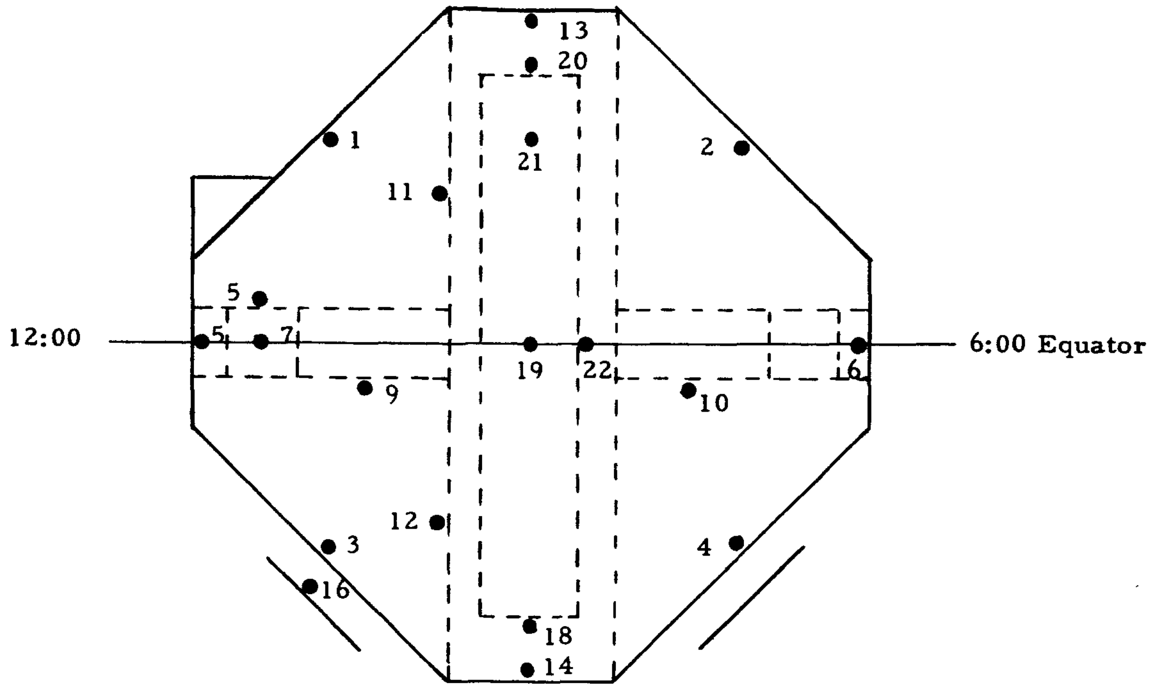


FIGURE 9. PROTOTYPE IN THE VACUUM CHAMBER

FIGURE 10. FOIL REFLECTOR PLACED OVER THE BLANKET

TABLE II. Thermocouples Locations



Location of Thermocouples

1. Midway of top cone at 12:00 on inside
2. Midway of top cone at 6:00 on inside
3. Midway of base cone at 12:00 on inside
4. Midway of base cone at 6:00 on inside
5. Inside equator at 12:00
6. Inside equator at 6:00
7. Inside battery pack at 12:00
8. On outside of same battery
9. On battery pack support midway between equator and center column at 12:00
10. Same as No. 9 except at 6:00
11. On outside of center shell midway between disk and bottom
12. On outside of center shell midway between disk and bottom
13. Inside top plate of instrument column
14. Inside base plate of instrument column
15. Outside of case of photoelectric cell
16. On underside of a micrometeorite sounding board
17. Midway of base cone at 9:00 on inside
18. On base of transmitter
19. Inside center of instrument column
20. Top center of instrument column
21. On base of commutator
22. Outside of instrument column opposite No. 19

Other Thermocouples Used

23. On cone of aluminum foil reflector (half-blanket test)
24. On end of aluminum foil reflector (half-blanket test)
25. On cone of LN_2 coils (half-blanket test)
26. On end of LN_2 coils (half-blanket test)

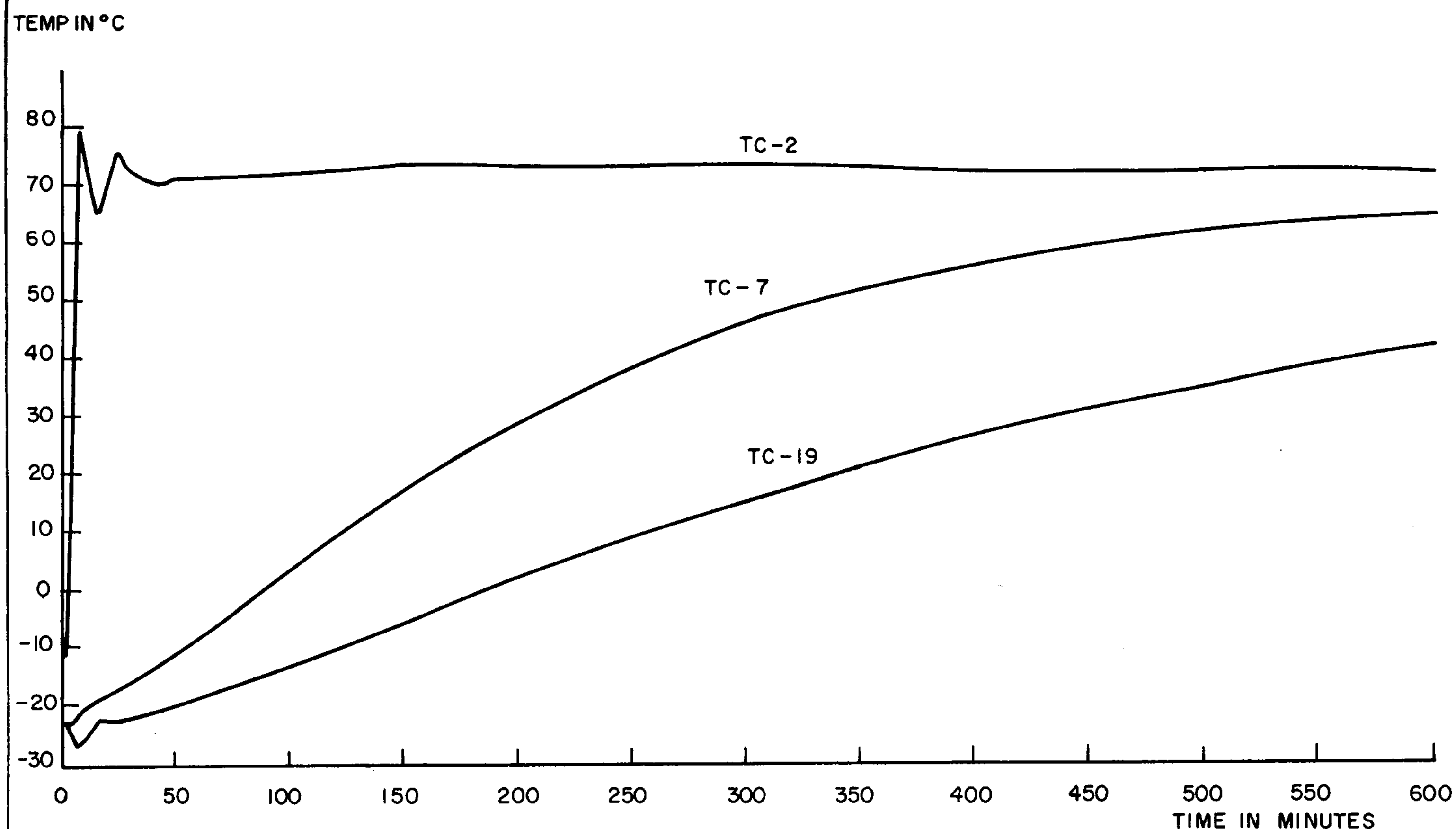


FIGURE 11. TEST NO. 164005 19-D WITH FULL BLANKET NO. 3

to the increase in temperature. As can be seen, the time constant (time for gradient to decrease by one-half) for the instrument column and the batteries is about six and two hours, respectively.

Results from one of the "half-blanket" tests (orbital simulation tests) are shown in Figures 12 and 13. Figure 13 demonstrates how the heat input from the blanket on the upper cone was regulated as the temperature of the cone changed. The temperatures of the payload and heat sink were monitored before applying the heat to the prototype, at which point the time scale on these figures is referenced. The near steady-state values reached at the end of about 5 hours running time (Fig.12) indicate the amount of heat transfer across the payload. Despite the large gradients across the payload it was found that due to the insulation there was only a relatively small gradient across the instrument column.

The results from these tests were fed into the computer program and it was found that no modifications were needed in the design of the payload from the thermal standpoint.

Blankets were used as the heat source instead of a type of solar radiation simulator; consequently, a check on the solar absorptivity of the surfaces during these tests could not be made. This was accomplished by making emissivity and reflectivity measurements on specially prepared samples. It was these measurements which indicated the need for the additional thermal coatings on the flight payloads.

RESULTS

The launch time of the Explorer VIII satellite was partly determined by thermal considerations. A time was selected when the percent of the orbit in sunlight and the attitude of the satellite with respect to the sun would remain as constant as possible throughout the anticipated two-month operational lifetime (Figs. 14 and 15). This would minimize the variations in the temperature of the sensitive components and keep the temperature near the center of the 0° to 50° C range. It was impossible to launch the payload so that the orbit would not move into a 100% sunlight situation at some time during the two months. Therefore, the launch time was chosen such that this condition would occur near the end of the expected lifetime. As it turned out, the batteries expired

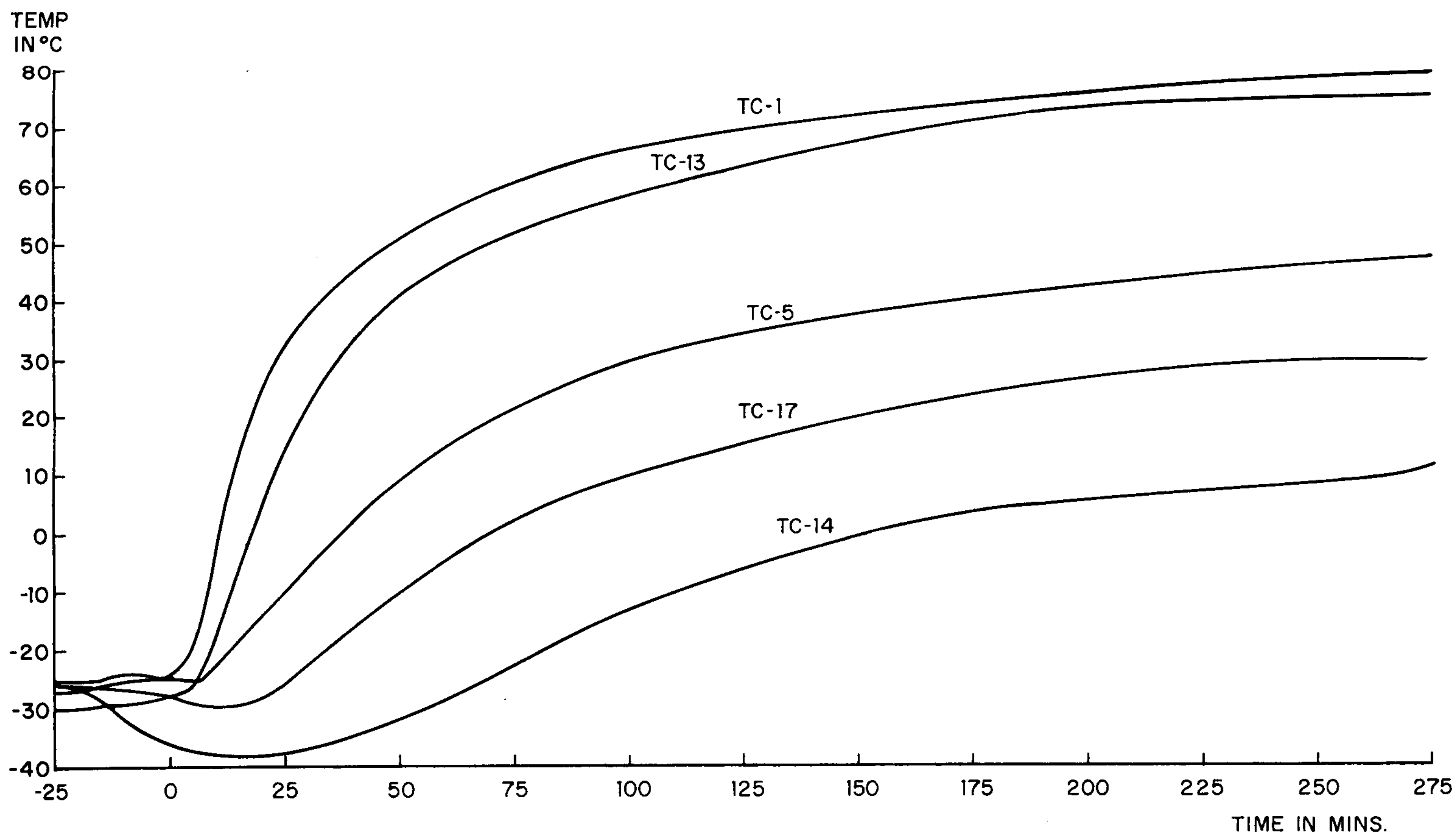


FIGURE 12. TEST NO. 164003 19-D WITH HALF-BLANKET NO. 1

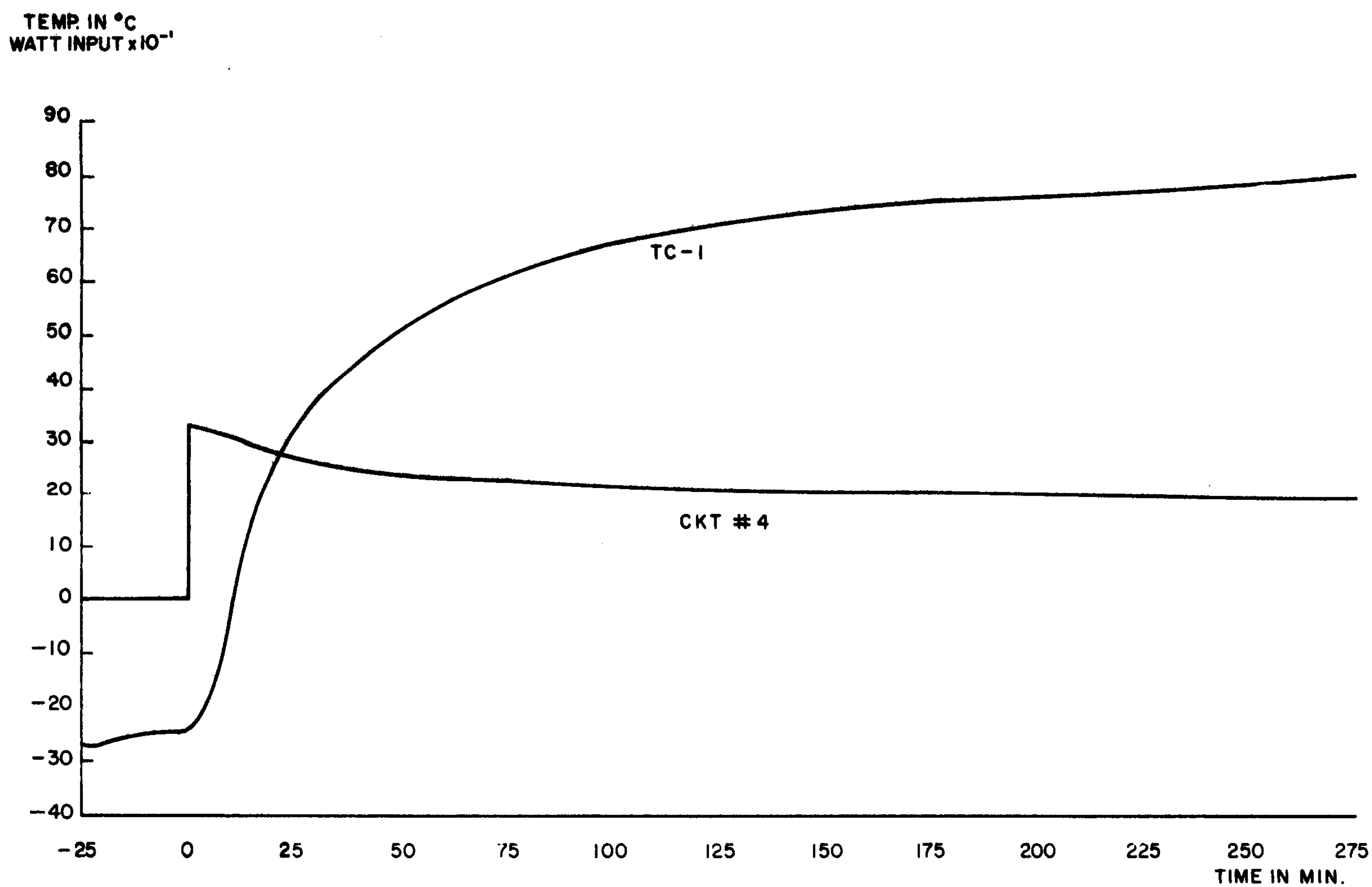


FIGURE 13. TEST NO. 164003 19-D WITH HALF-BLANKET NO. 1

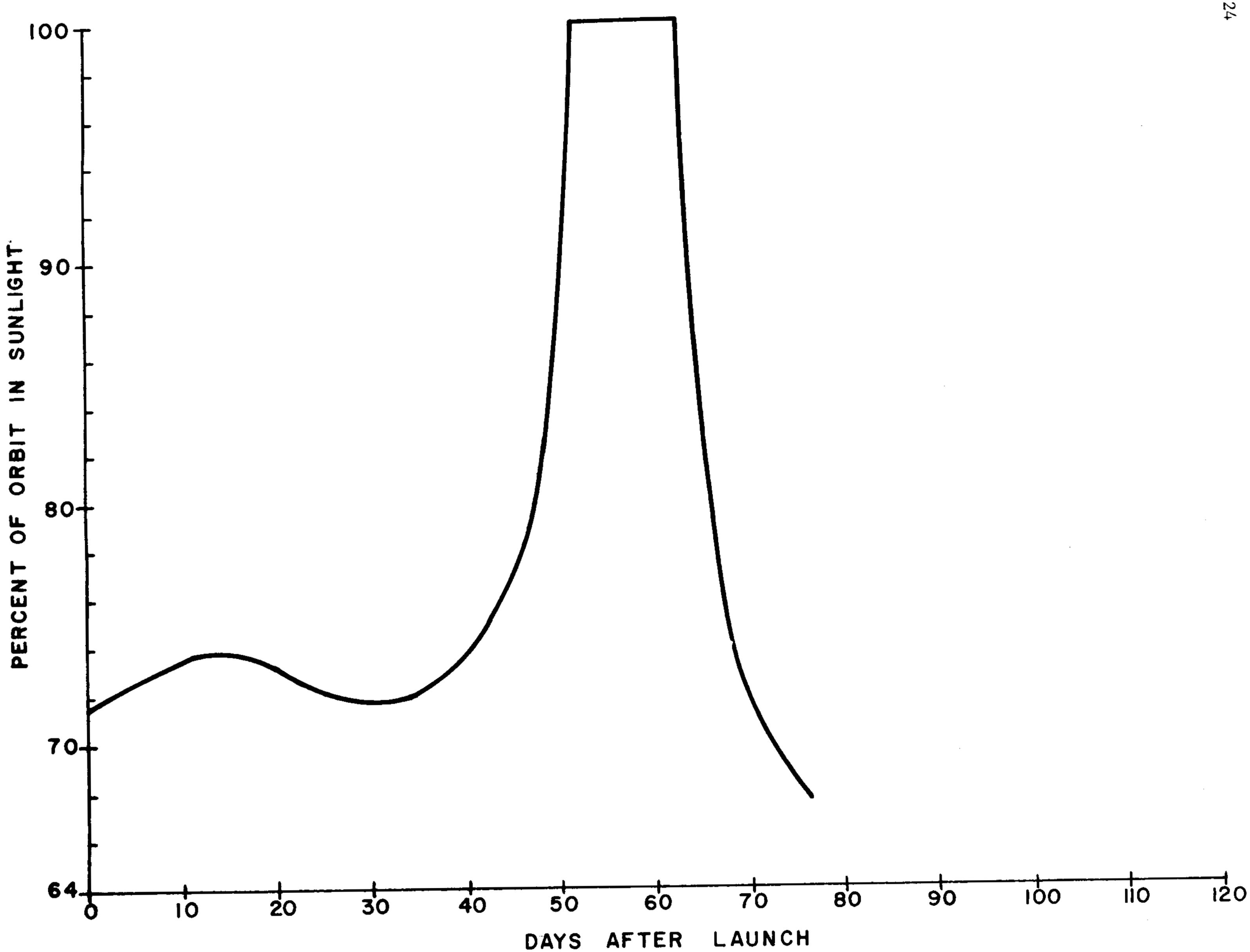


FIGURE 14. PERCENT OF ORBIT IN SUNLIGHT VS DAYS AFTER LAUNCH

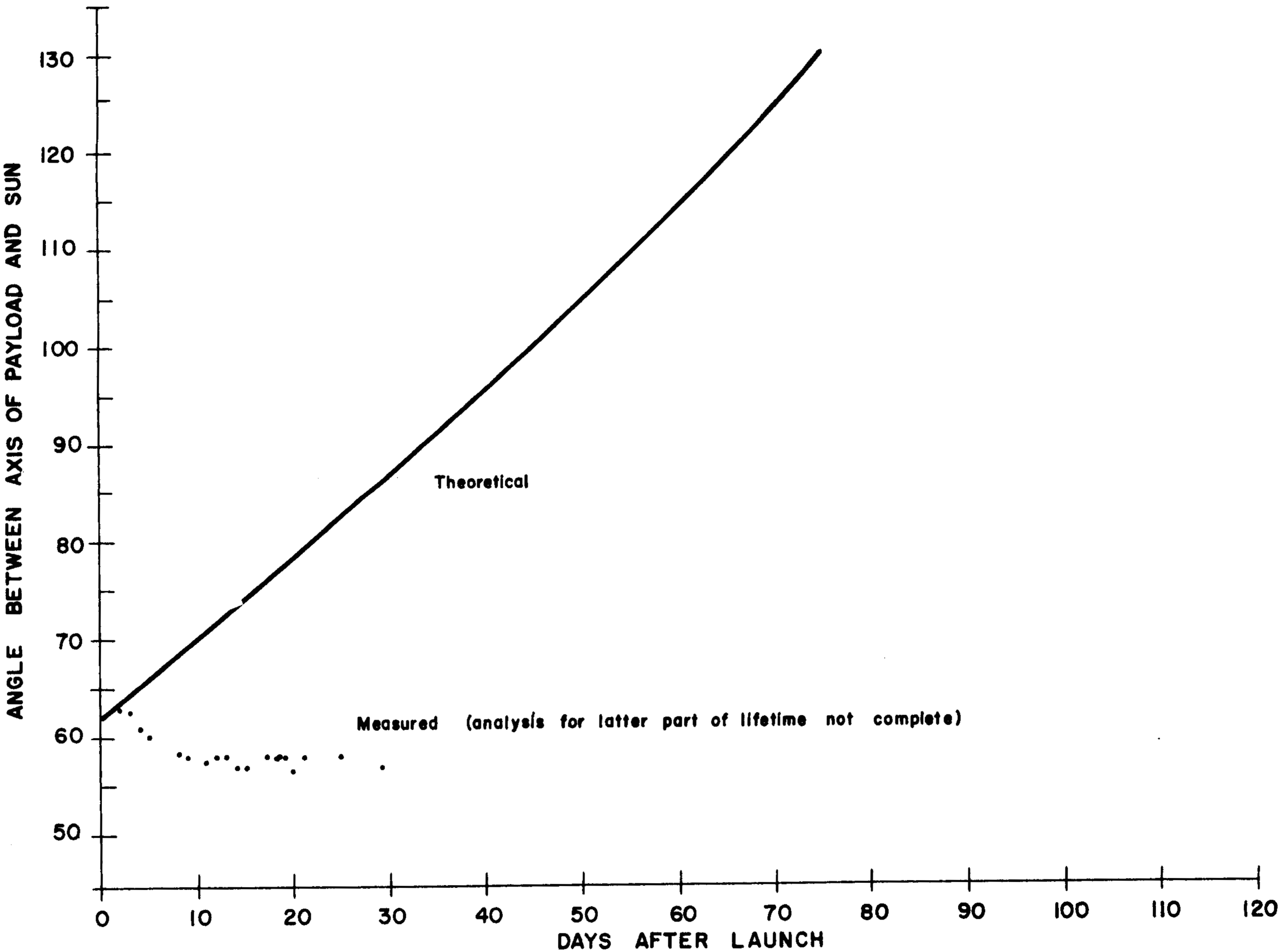


FIGURE 15. ALTITUDE OF PAYLOAD VS DAYS AFTER LAUNCH

before this situation occurred. As reported by Bordeau, Donley, and Whipple [1], the interaction between the payload and the earth's magnetic field caused deviations in the solar aspect angle from the predicted values; this, however, caused no thermal problems. The temperature of the satellite was monitored by means of thermistors at four locations: (1) the center of the instrument column, (2) battery pack, (3) equatorial rim, and (4) meteorite photomultiplier. The temperatures of the instrument column and batteries are shown in Figure 16. It can be seen that the instrument column remained within the range of 22°C to 33°C and the batteries within the range of 17° to 27°C.

The temperatures of the equatorial rim and the meteorite photomultiplier during part of a revolution of the satellite about the earth are shown in Figure 17. This plot was obtained by using temperature measurements over a period of several days and plotting them on the same graph as a function of the position of the satellite with respect to the ingress into the earth's shadow. It was possible to place the measurements on the same plot because there were only small changes in the percent of the orbit in sunlight and the attitude with respect to the sun during this time. This graph shows that the temperature of the rim, being part of the skin of the satellite, fluctuated over a range of about 40°C during each revolution, and the photomultiplier over a range of about 25° C. A prelaunch theoretical curve of the rim temperature is also shown. The slight deviations between the theoretical curve and the measured data is probably due to the theoretical value used for the heat capacity of the skin, emissivity of the skin, the thermal coupling between areas, or some combination of these factors. It is thought that extensive analysis of the data should yield information as to the exact cause of this slight difference; however, the time and effort involved make this analysis impractical for the present.

CONCLUSIONS

It is felt that the thermal design of the S-30 payload was successful since the temperatures of the thermally sensitive components remained within the allowable range. The temperature measurements made on the satellite indicate that the philosophy of controlling the average temperature of the skin by a selection of proper coatings and reducing the temperature fluctuations of the sensitive components by the use of insulation

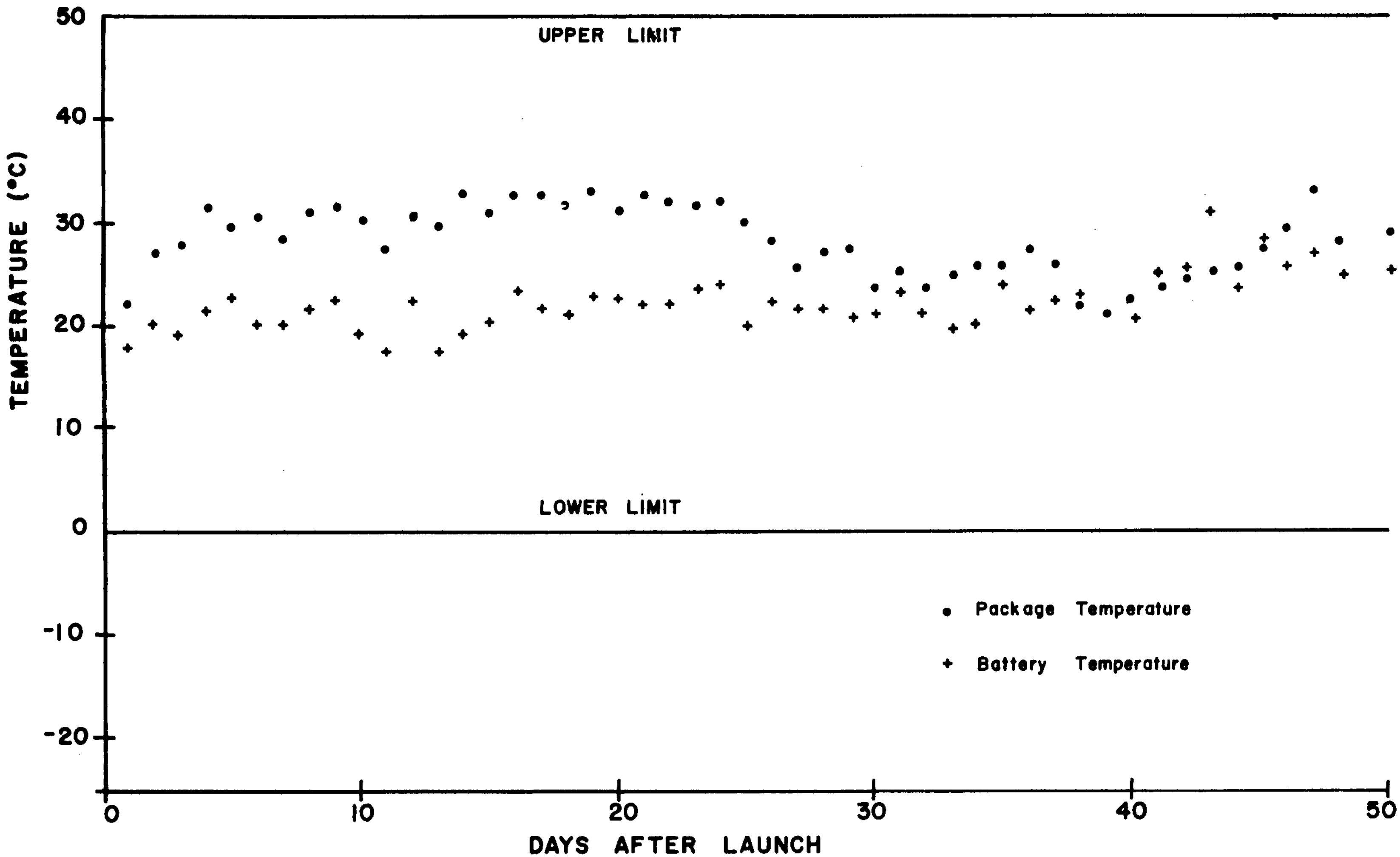


FIGURE 16. INSTRUMENTAL PACKAGE AND BATTERY TEMPERATURE VS DAYS AFTER LAUNCH

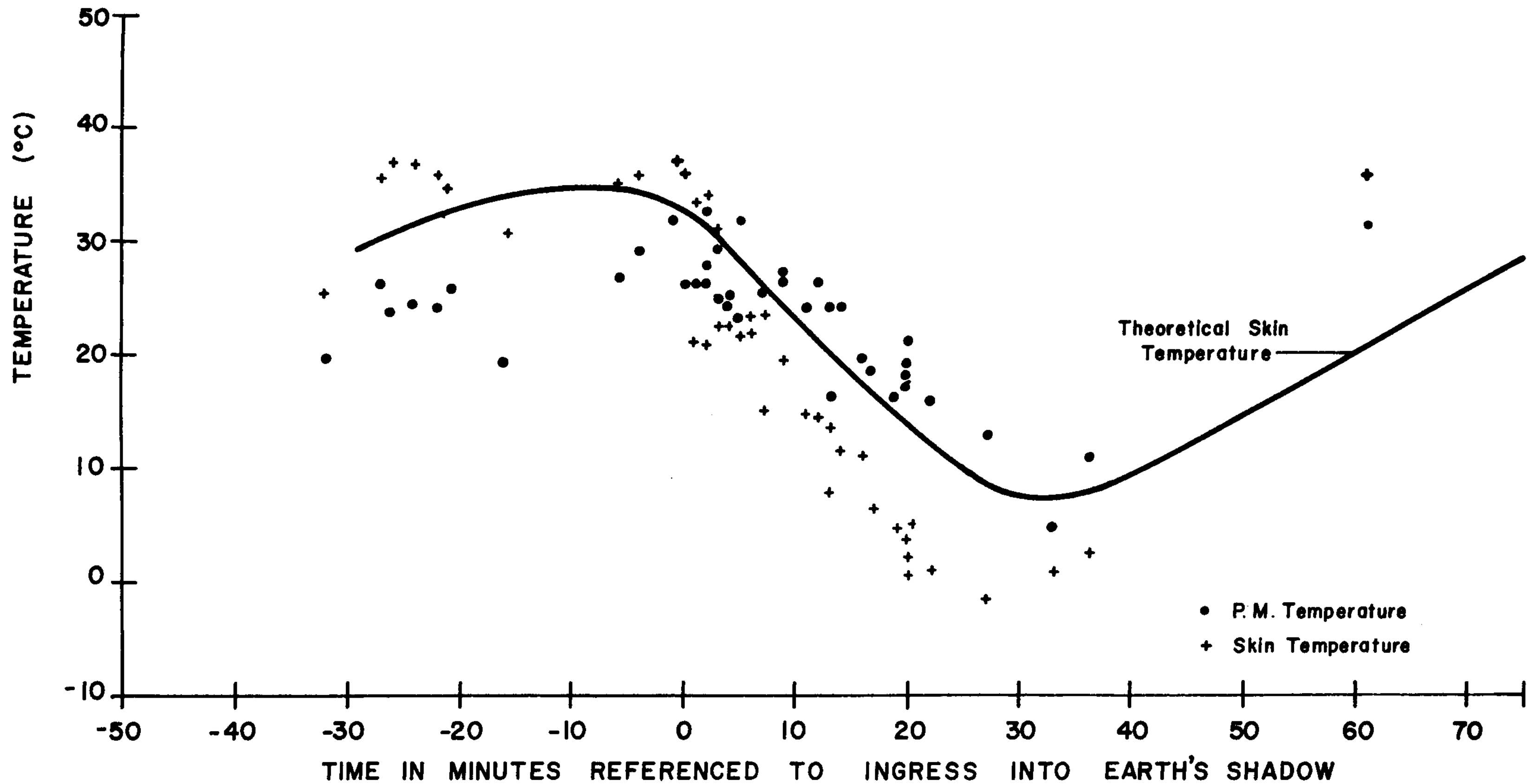


FIGURE 17. TEMPERATURE OF EQUATORIAL RIM AND METEORITE PHOTOMULTIPLES DURNIG REVOLUTION

was satisfactory for meeting the thermal requirements imposed upon the payload design.

The small number of temperature measurements on the payload make it impossible to determine how well the paints held up in the space environment. It is possible that changes might have occurred which had a cancelling effect. Thus, it can only be stated that if any major changes took place, the net effect on the thermal design was small.

The comparison between theoretical and measured temperatures gave a good check on the theoretical calculations and the approach used in making them. The agreement obtained gives a higher level of confidence to the method used.

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